N90-28637

HIGH/VARIABLE MIXTURE RATIO OXYGEN/HYDROGEN ENGINES

Wm. H. Knuth

and

John H. Beveridge

Acurex Corporation-Huntsville Operations
Aerotherm Division
1500 Perimeter Parkway, Suite 225
Huntsville, AL 35806
(205) 830-1230

ABSTRACT

A LOX/LH_2 high/variable mixture ratio booster-upper stage engine is described. The engine has high thrust-weight ratio as a booster and high specific impulse as an upper stage engine. Operation at high mixture ratio utilizes the propellants at high bulk density.

The engine may use multiple turbopump-preburners for higher thrust ratings. The engine uses the full flow cycle to obtain minimum turbine inlet temperatures for a given chamber pressure and to avoid interpropellant shaft seals and other single point failure modes.

A portion of the liquid hydrogen is used to regeratively cool the thrust chamber assembly. The warmed hydrogen coolant is then used to drive the fuel boost turbopump.

All propellants arrive at the gas-gas injector ready to burn. Shear mixing of the parallel flowing high velocity, low density fuel-rich gases with the high density, low velocity oxidizer-rich gases provides complete combustion with a modest chamber volume. Combustion stability is assured by the injection of the heated fuel-rich gases and the comparatively low volume ratio of the propellants before and after combustion.

The high area ratio nozzle skirt is fitted with a low area ratio nozzle skirt insert for optimum low altitude performance. The overall engine characteristics make it a candidate for ALS, Shuttle-C, LRB and SSTO applications.

INTRODUCTION The study of the high variable/mixture LOX/LH_2 engine concept was begun at Acurex (Cryomec Propulsion) about four years ago in conjunction with the full flow, staged combustion cycle studies. The high mixture ratio feature was intended for booster engine application wherein reduced engine specific impulse was traded for increased bulk density of the propellants. The result of this trade was to make the LOX/LH_2 propellant combination more attractive for booster propulsion and thereby make a single engine design and a single propellant combination attractive for earth-to-orbit rocket propulsion. This approach has great merit for cost reduction of payload-to-orbit.

In the course of our studies we were referred to the 1977 work of J.A. Lombardo and D.H. Blount of the MSFC. They optimized vehicle performance by taking advantage of the nonliner characteristics of the specific impulse vs bulk density curve of LOX/LH_2. They showed the desirability of high mixture ratio for the early portion of burn; and low mixture ratio, maximum specific impulse for the late portion of the powered flight.

VARIABLE MIXTURE RATIO ENGINE The basic engine has a chamber pressure of 2250 psia, operates at a mixture ratio of 6:1, has a vacuum thrust of 523K lb with an expansion area ratio of 64 and a specific impulse of 453 seconds. Variants of the design would add a second LOX turbopump which would increase mixture ratio to 9:1 and sea level thrust to approximately 643K lb. The chamber pressure would increase to 3000 psia. This latter engine variant is designed to serve both as the booster and the sustainer engines of a space launch vehicle.

For still higher thrust requirements, engines using multiple turbopumps (up to four LOX turbopumps and two LH_2 turbopumps) are further potential outgrowths of this concept. At 1500K lb sea level thrust, such engines maintain extensive component commonality with the smaller thrust versions, while providing thrust levels and component redundancy desired for very large launch vehicles.

The technology base for the oxidizer-rich LOX/LH_2 preburner and the gas-gas main injector used in the engine have been previously investigated. NASA-MSFC has run a successful LOX/LH_2 preburner test series over a wide range of oxygen-rich operation (mixture ratios from 20-150). NASA LeRC, (and others) have run gaseous oxygen/gaseous hydrogen-fed combustion chambers successfully. The engine defined herein has no known new or advanced technology areas.

It is axiomatic that an advanced propulsion system use existing proven propellant combinations if it is to have a low risk, low cost development program. Three such propellant combinations predominate. They are LOX/LH_2, LOX/RP-1 and N_2O_4/50-50 Hydrazine/UDMH. The toxicity of the latter combination rules it out for the next generation launch vehicles because of environmental considerations. The performance of LOX/RP-1 is limited by heat transfer and cooling considerations. LOX/Hydrocarbon variants such as LOX/propane or LOX/methane depart from established experience. There are no large LOX/propane or LOX/methane engines in the world. Technology programs have also disclosed potential limitations with coking and combustion stability with some of the hydrocarbons.

A means to extend the performance of LOX/RP-1 by adding LH_2 as the coolant and turbine drive fluid i.e., a tripropellant engine, has been studied. Results suggest a low dry weight for the tripropellant vehicle. However, the complexity of the tripropellant engine, tripropellant vehicle and tripropellant ground support equipment all portend high development cost (and risk) as well as high operational costs. The complexity of the tripropellant engine also makes it unlikely that it can be derived from an existing engine. It does not appear to be a viable candidate for a low cost space transport system.

LOX/LH_2 is the remaining proven propellant combination. Used in the RL-10, J-2, and SSME, this propellant combination offers the features of high performance, clean exhaust and lowest vehicle gross lift-off weight (GLOW). When mixture ratio is varied during the launch profile in an optimized schedule, a LOX/LH_2 vehicle's size and dry weight become directly competitive with other propellant combinations.

Lowest vehicle GLOW is important for low cost per pound to low earth orbit (L.E.O). A typical comparison would show a LOX/LH_2 vehicle being 20%-30% lighter at lift-off than a LOX/hydrocarbon vehicle for the same mission. This translates to a direct savings of 20%-30% in propulsion system cost. And since the majority of the lift-off weight differential is in propellant mass, the LOX/LH_2 system does not have to throttle as deeply as the LOX/hydrocarbon propulsion system, thereby suggesting a more simple design and reduced demand on controllability.

Recent STAS study results have determined that booster engines on the order of 650K-750K lb thrust are of the right size category. Sustainer engines of about 50%-60% of booster engine thrust are also perceived as the right size

range. These engine sizes provide reasonable engine-out capability, and are yet large enough to cluster within a conventional vehicle base area. This engine size also is consistent with STAS results identifying the optimum vehicle in the smaller size ranges to avoid excessive payload value per launch.

DESCRIPTION OF THE HIGH VARIABLE MIXTURE RATIO DERIVATIVE ENGINE

LOX/TURBOPUMP The derivative engine is derived from SSME technology by converting the LOX turbopump preburner to operate oxidizer-rich as schematically illustrated in Figure 1. With all the propellant mass flow then available as turbine drive fluid, turbine operating temperatures can be reduced. Also there is oxidizer-rich gas drive fluid available to drive the low pressure LOX boost pump. More boost pressure (an additional 100 psi) can be provided and LOX recirculation of up to 20% back to the boost turbopump turbine is no longer needed. This together with the use of cool, dense, oxidizer-rich gas to drive the main LOX turbopump turbine, results in major design simplifications. Figure 2 shows a comparison of the LOX turbopumps for the SSME and the derivative engine.

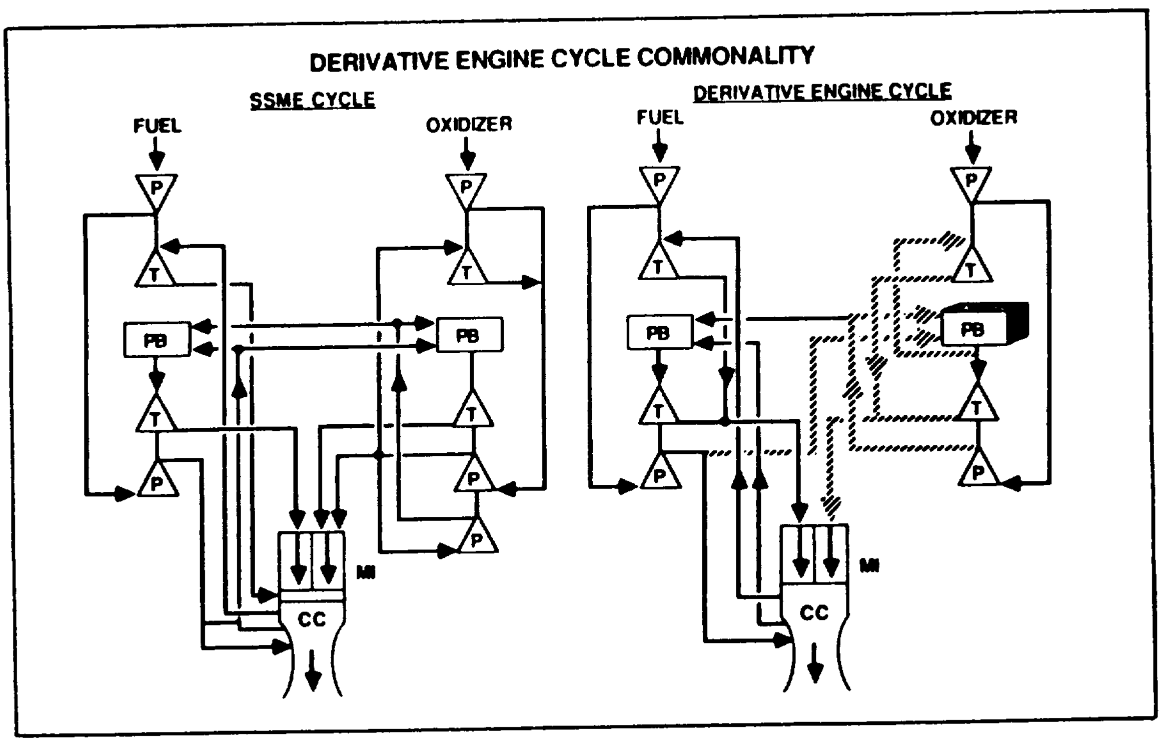

Figure 1: Derivative Engine Cycle Schematic

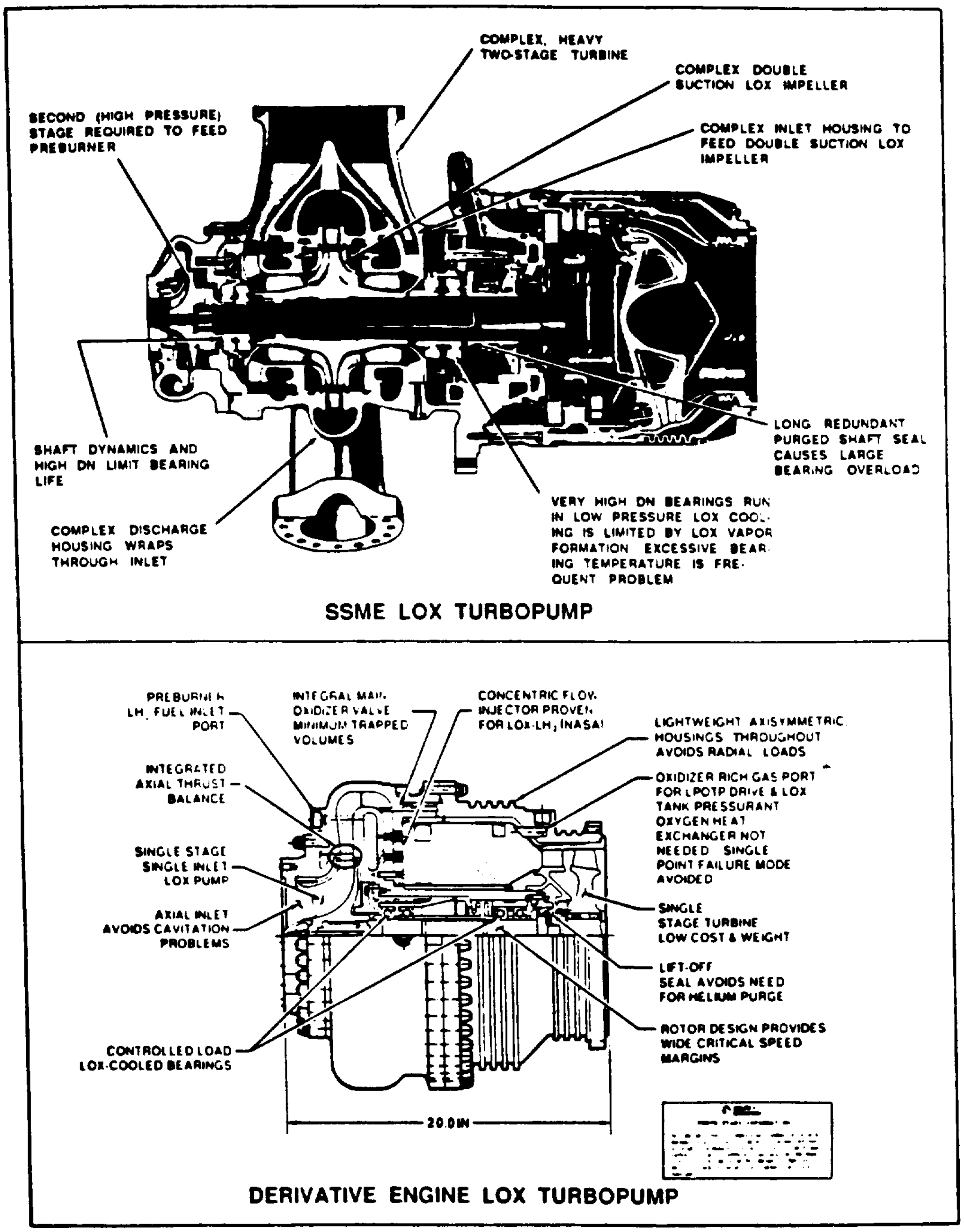

Figure 2: Comparison Of The SSME And Derivative Engine LOX Turbopumps

ORIGINAL PAGE IS OF POOR QUALITY

LH_2 TURBOPUMP Essentially all of the hydrogen flow becomes available to drive the LH_2 turbopump turbines. The main turbopump turbine inlet temperature is reduced from 1500°R to 800°R for a chamber pressure of 3000 psia. Figure 3 shows turbine inlet temperatures for 2000 and 3000 psia chamber pressure as a function of mixture ratio. Note the modest temperatures. Lower temperature capability turbine materials, such as A286, can be used which are insensitive to hydrogen embrittlement. The current SSME practice of gold plating the turbine components can be avoided. At the lower temperature and greater mass flow, the volumetric flow through the LH_2 turbopump turbine remains about the same as before. Therefore, it is possible to use the LH_2 turbopump essentially as is. Figure 4 illustrates the LH_2 turbopump. A simplified (low cost) design for an expendable application of the derivative engine is also shown.

GAS DUCTING The hot gas manifold (HGM) of the SSME is fitted with a liner and hydrogen cooled. It is physically very complex as illustrated by Figure 5. The HGM has also been the focus of extensive computational fluid dynamics to resolve problems of flow instabilities and excessive pressure drop.

The gas ducting of the derivative engine does not require cooling, and so is unlined. The flow area is larger and better streamlined without the liner, and the duct receives gas flow only from both the LH_2 turbine and the LOX turbine exhaust. The net result is a duct with more uniform flow and temperature distribution.

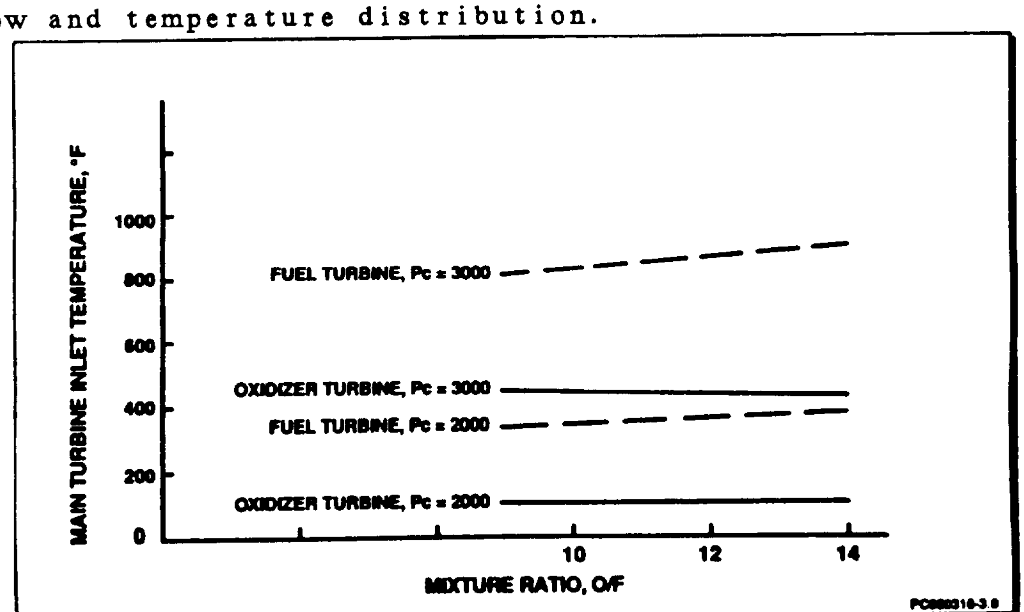

Figure 3: Turbine Inlet Gas Temperature Versus Mixture Ratio And Chamber Pressure

ORIGINAL PAGE IS
OF POOR QUALITY

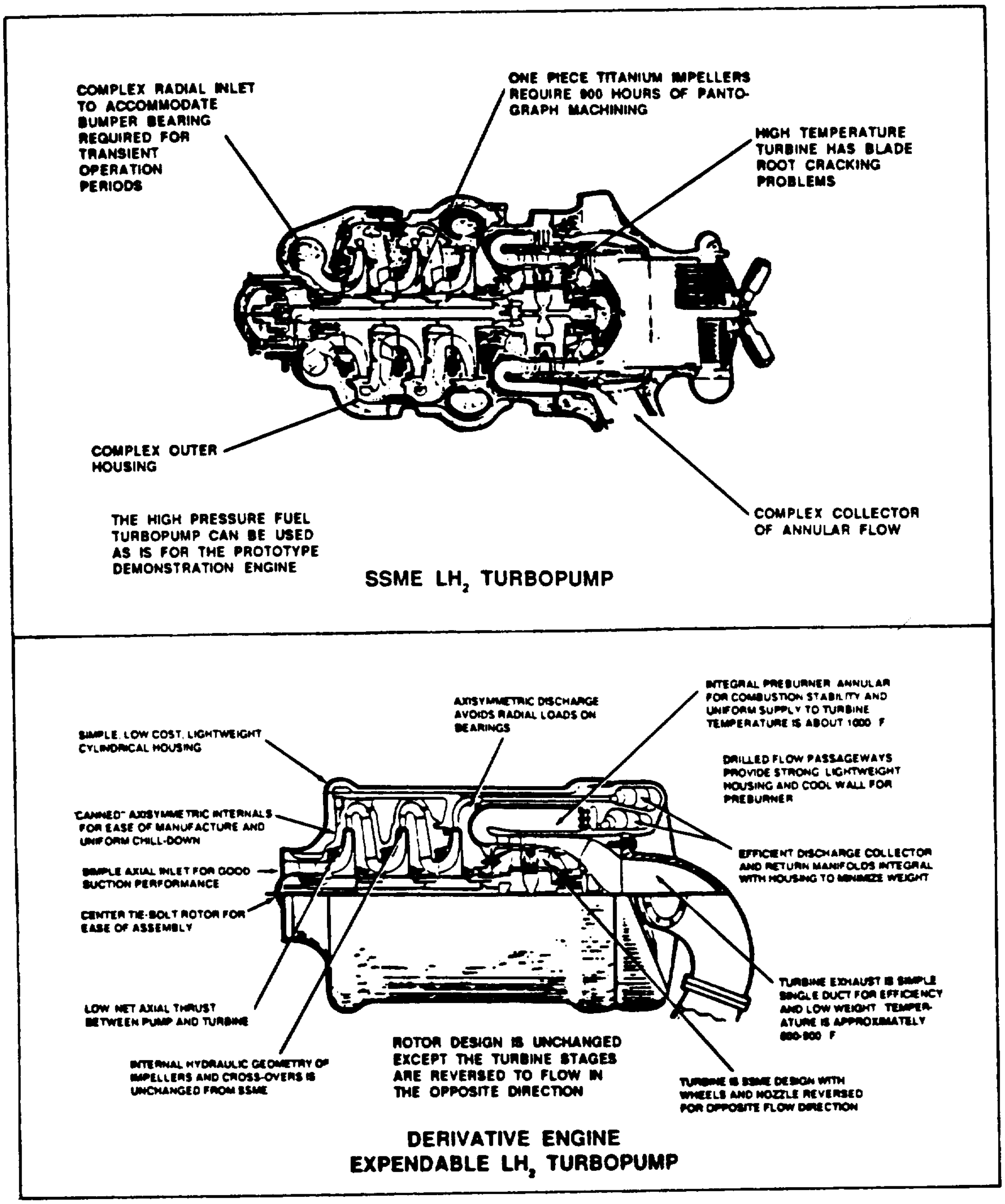

Figure 4: The SSME & Derivative Engine LH_2 Turbopumps

ORIGINAL PAGE IS
OF POOR QUALITY

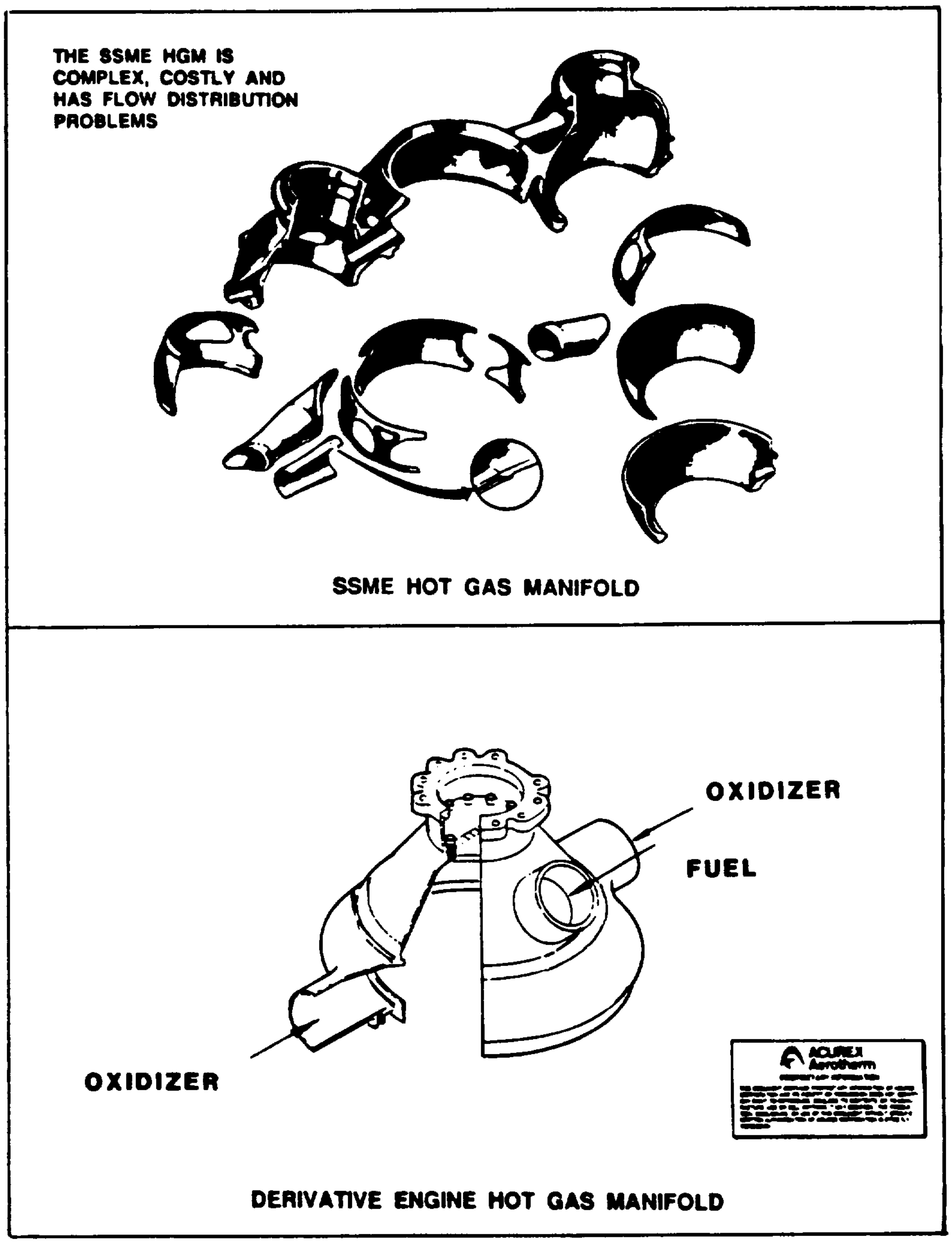

Figure 5: Hot Gas Manifold Comparison

MAIN INJECTOR The main injector of the current SSME delivers hot fuel-rich gas from the turbine exhausts and LOX from the high pressure LOX turbopump, plus some residual GH_2 to the main chamber. The present injector needed to successfully inject these three widely different fluids is complex as illustrated in Figure 6.

The main injector for the derivative engine needs only to inject two compressible fluids received at similar temperatures and pressures. The injector for such service is not complex, but rather composed of simple divider plates. As illustrated in Figure 7, the divider plates may be in the shape of vanes, as shown in contrast to the current SSME injector. Other gas-gas injector geometries may be found to be more desirable, as shown in Figure 7. This injector has an array of tubes for injection of the fuel-rich gases and a rigimesh injector face for the injection of the slow moving oxidizer-rich gases. The rigimesh ensures a flat distribution of oxidizer-rich gases across the face of the injector. The high velocity, low density jets of fuel-rich gases rapidly decay and mix for rapid complete combustion with the dense gaseous oxygen.

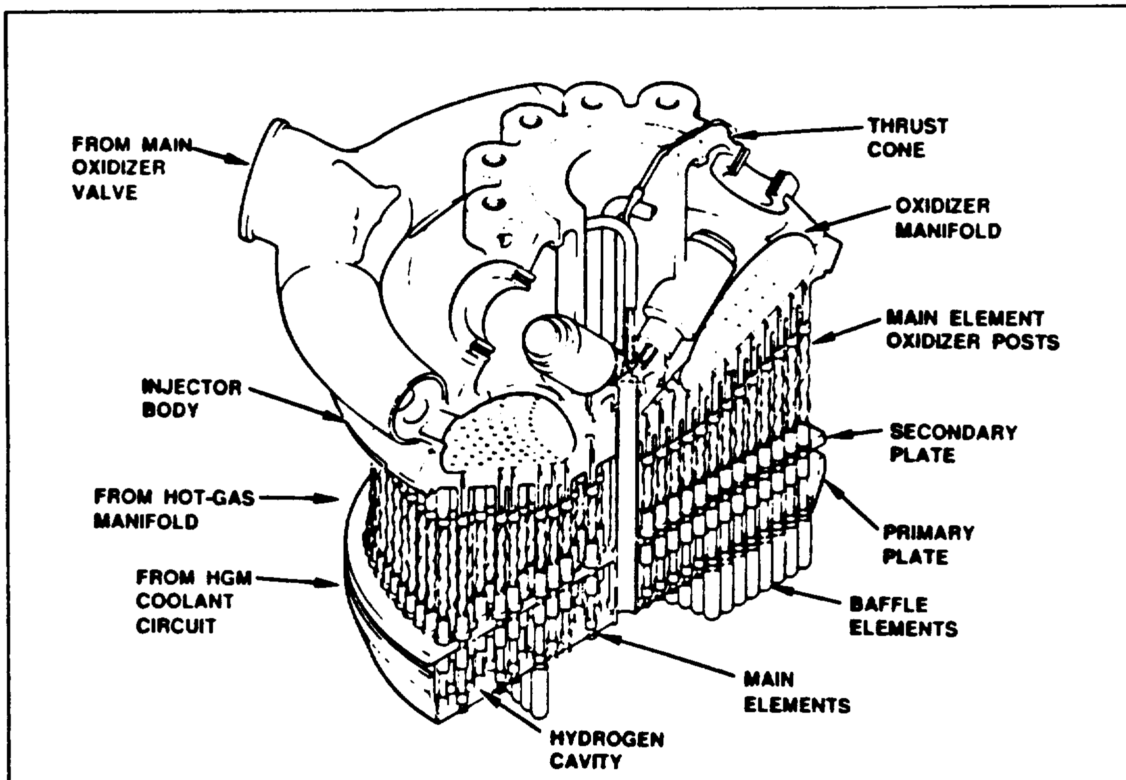

Figure 6: SSME Main Injector

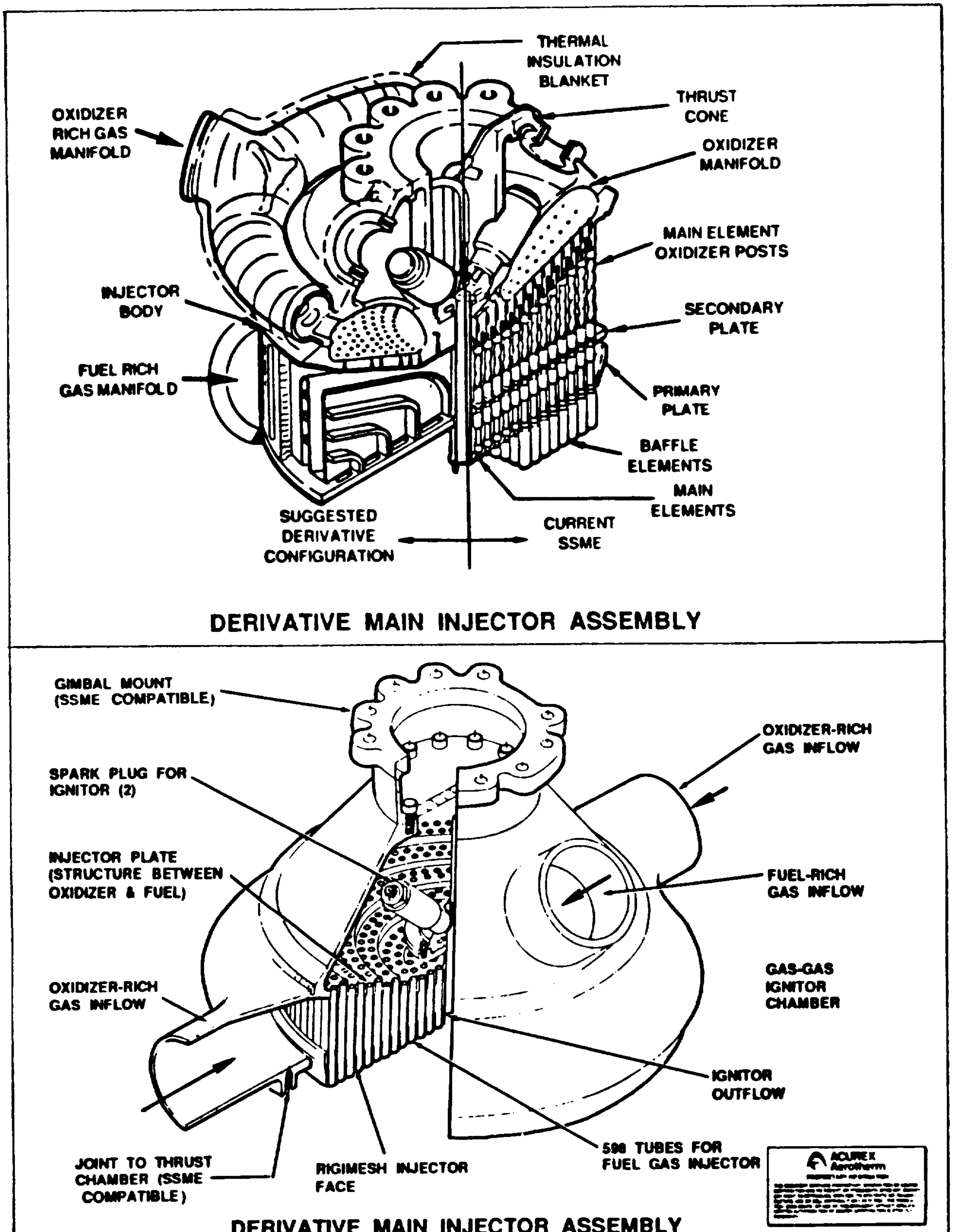

Figure 7: Derivative Main Injectors

COMBUSTION PROCESSES Partial combustion of the propellants occurs in the preburners of the full-flow cycle LOX/LH_2 engine. The bulk volume increase of the propellants due to partial combustion is a volume ratio factor of about 5:1. When these gaseous propellants are injected via the gas-gas injector into the main thrust chamber and burned, an additional increase in average bulk specific volume occurs, i.e., again a ratio of about 5:1. This same ratio is on the order of 75:1 for liquid injection LOX/hydrocarbon thrust chambers. The rapid and large change in bulk specific volume of the propellants during combustion in a liquid-fed chamber is the prime cause or driver for combustion instability. As a result of dividing the overall combustion process into two stages, (i.e., the full flow cycle) and the use of LOX/LH_2 propellant combination, potential for combustion instability is inherently minimized. As a consequence relatively simple main injectors can be utilized.

The mixing and combustion of the oxidizer and fuel gas streams is easy to model with the computer. No recirculation of combustion products back to the injector face is necessary to vaporize the propellants; they are combustion ready at injection and mixing.

PREDECESSOR/DERIVATIVE SIMILARITY The engine resulting from this approach has a generic similarity to the SSME. The good features of SSME are retained, while advantages and simplifications of the new design are incorporated. Figure 8 shows a schematic comparison of the SSME and derivative engine power heads.

The simplifications achieved by the derivative engine will reflect directly in lower cost, higher safety and reliability, lower weight and wider operating margins. Virtually all these benefits can be demonstrated by a test bed engine assembled from end-of-life SSME components. Figure 9 illustrates the commonality with SSME hardware.

BOOSTER-SUSTAINER EVOLUTION The basic derivative LOX/LH_2 engine outlined in the preceeding sections is intended as a high performance, low cost, reuseable or expendable engine with a vacuum thrust rating of approximately 523K lb. It is also useable as a direct substitute for SSME.

Its development also provides the component building blocks which can be used to produce a high thrust variable mixture ratio engine. Such an engine serves as a high mixture ratio booster at liftoff and transitions to a lower mixture ratio, high Isp sustainer mode at an appropriate time in the flight. This is done by adding a second LOX turbopump, identical to the first, to the engine and increasing

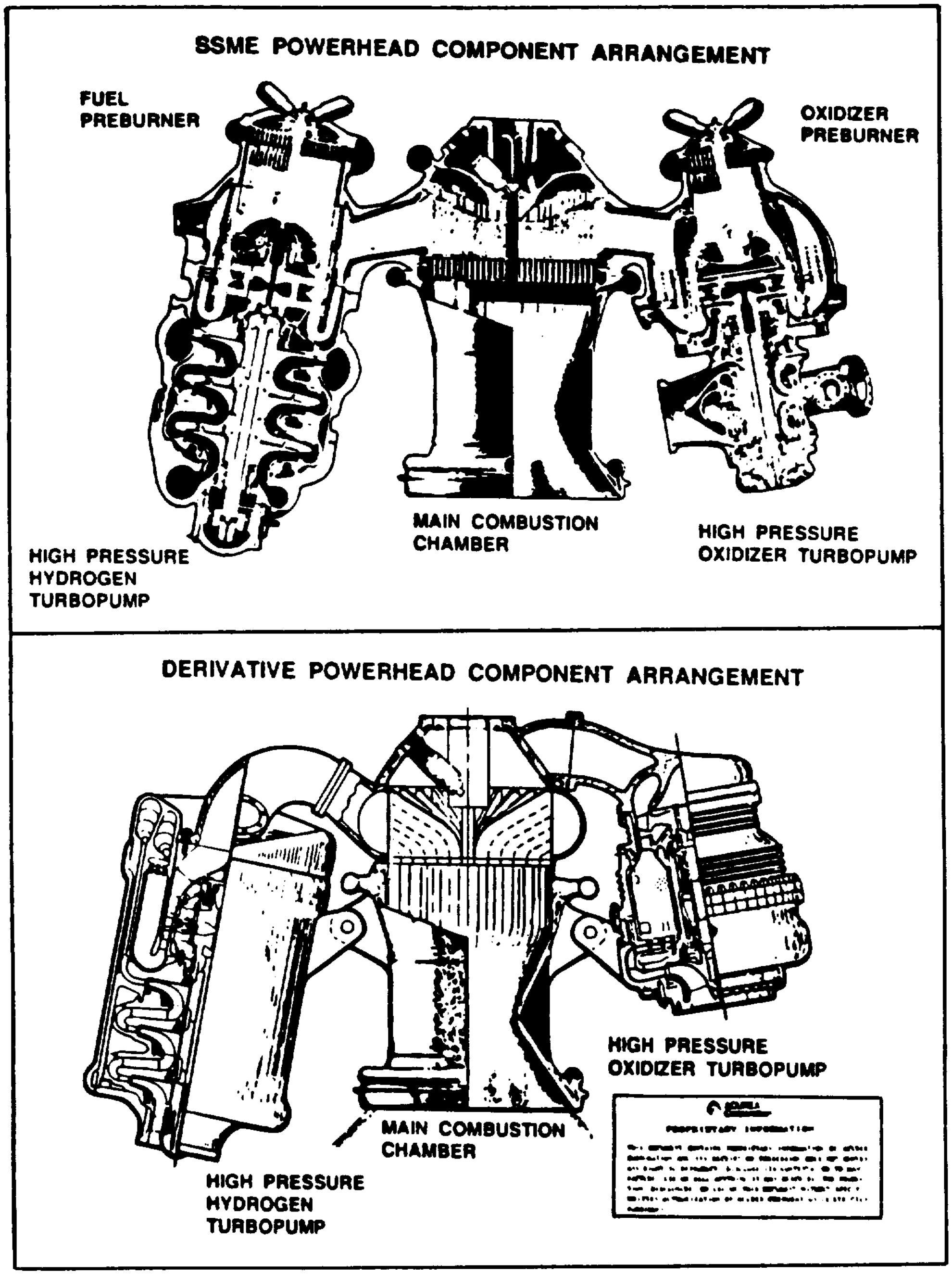

Figure 8: Comparison Of The SSME And Derivative Engine Power Heads

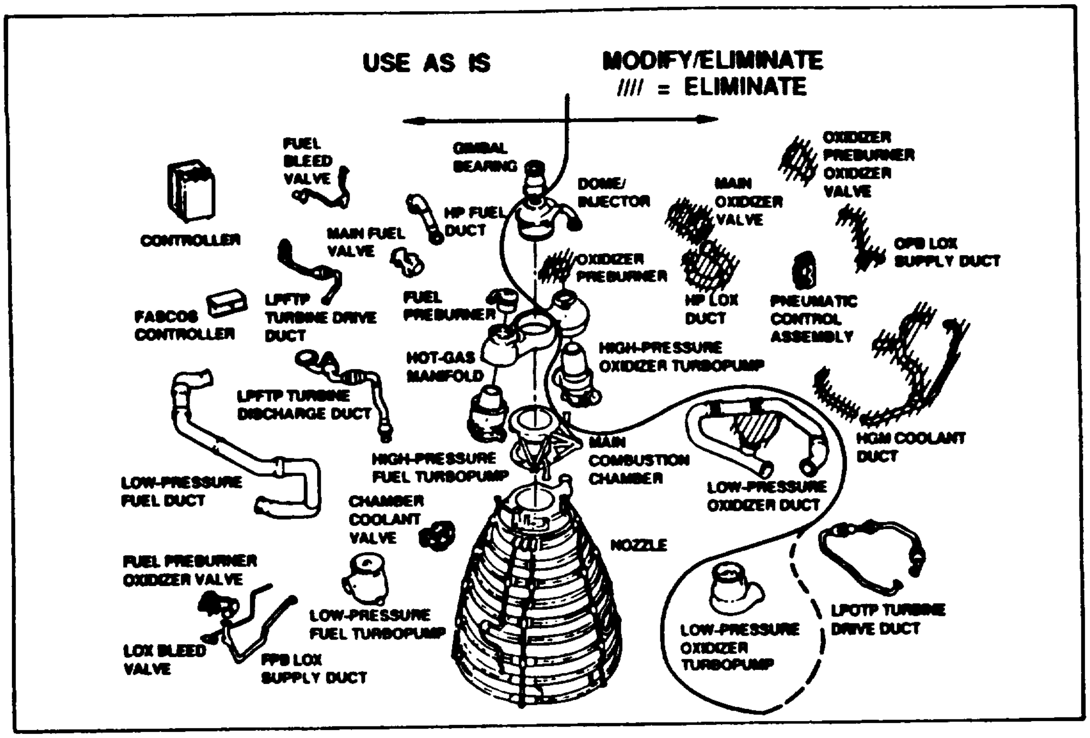

Figure 9: SSME Major Components For Derivative Test Bed Engine

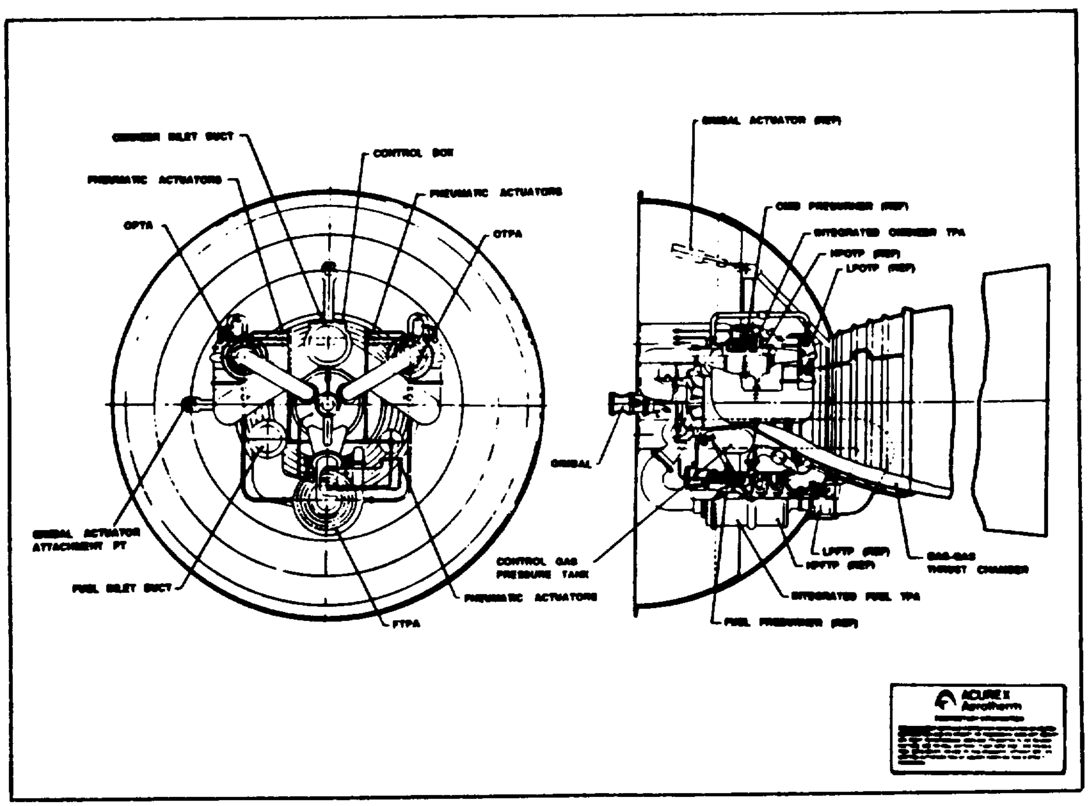

Figure 10: Variable Mixture Ratio LOX/LH_2 Engine

the nozzle throat area accordingly. Figure 10 illustrates the variable mixture ratio derivative engine with three main turbopumps. Table 1 and Table 2 list characteristics in the booster and sustainer engines.

Addition of the second LOX turbopump increases engine vacuum thrust by nearly 35% while increasing engine weight and cost by only about 15%. This engine thrust at sea level is 643K lb_f. Continuing this strategy, six turbopumps, four LOX and two LH_2 can be used on a single chamber. This larger engine, illustrated in Figure 11, provides thrust over 1500K lb_f. Mixture ratio is changed by shutting off LOX turbopumps at such time as it is desired to reduce thrust to maintain acceleration limits. Thrust in the sustainer mode can ultimately be reduced to the equivalent of 1/4 the original thrust by operating one LOX turbopump and one LH_2 turbopump. The turbopumps can also be throttled back for still further thrust reduction as required for single stage-to-orbit vehicles.

The multiple turbopump engine provides turbopump redundancy. The engine can operate safely and satisfactorily even if a safety system shuts down one or more turbopumps. It is pertinent to note that a multiplicity of turbopumps on a single chamber provide a higher degree of redundancy than the same number of turbopumps used in the conventional manner of one oxidizer pump and one fuel pump per chamber. The reason for this is that a turbopump failure on the conventional engine forces the loss of its companion pump and related thrust chamber. On the multi-turbopump engine, the failed turbopump can be shut down, while all other turbopumps continue to operate. Thrust loss is minimal, and can be avoided by accelerating the remaining turbopumps.

The foregoing discussion was limited to oxygen and hydrogen. Use of oxygen-rich combustion to achieve high bulk density of the propellants is not limited to the use of oxygen as diluent. Figure 12 shows vacuum specific impulse versus bulk specific volume for several other diluents. These diluents can be a fuel or an inert fluid. The advantage of an inert fluid is that the products of combustion would remain fuel-rich rather than oxidizer-rich. Using LOX as a baseline, the addition of water, nitrogen or ammonia decrease the system performance while the addition of the monopropellant fuel hydrazine increases the system performance. Use of such other diluents adds the complexity of a third propellant system to the vehicle and to the engine. The general conclusion concerning alternative diluents is that, unless oxygen-rich combustion demonstrates a propensity to "burn up" the thrust chambers, LOX is the best diluent choice.

TABLE 1: BOOSTER

AREA RATIO	20	64
THRUST (VAC), lbs	679,952	700,000
MIXTURE RATIO, o/f	9	9
CHAMBER PRESSURE, psia	3000	3000
AREA THROAT, sq in.	125.6	125.6
DIAMETER THROAT, in.	12.64	12.64
DIAMETER EXIT, in.	56.6	101.2
WEIGHT FLOW RATE, OXIDIZER, lb/sec	1485	1485
WEIGHT FLOW RATE, FUEL, lb/sec	165	165
TOTAL WEIGHT FLOW, lb/sec	1650	1650
SPECIFIC IMPULSE (VAC), sec	412	424
ENGINE DRY WEIGHT WITH NSI, lbs	6860	---
ENGINE DRY WEIGHT WITHOUT NSI, lbs	---	6565
ENGINE THRUST-TO-WEIGHT RATIO	99	107
ENGINE LENGTH, in.	154	154
DIAMETER POWER HEAD, in.	100	100
THRUST (S.L.) lbs	642,966	---
SPECIFIC IMPULSE, (S.L.), sec	390	---
PRESSURE PUMP DISCHARGE, psia	6400	6400
TURBINE PRESSURE RATIO	1.58	1.58
TURBINE INLET TEMPERATURE, ^{o}F		
OXIDIZER	430	430
FUEL	830	830
SHAFT SPEED, rpm		
OXIDIZER	17,200	17,200
FUEL	34,400	34,400

TABLE 2: SUSTAINER ENGINE

AREA RATIO	64
THRUST (VAC), lbs	523,225
MIXTURE RATIO, o/f	6
CHAMBER PRESSURE, psia	2250
AREA THROAT, sq in.	125.6
DIAMETER THROAT, in.	12.64
DIAMETER EXIT, in.	101.2
WEIGHT FLOW RATE, OXIDIZER, lb/sec	990
WEIGHT FLOW RATE, FUEL, lb/sec	165
TOTAL WEIGHT FLOW, lb/sec	1155
SPECIFIC IMPULSE (VAC), sec	453
ENGINE DRY WEIGHT, WITHOUT NSI, lbs	6565
ENGINE THRUST-TO-WEIGHT RATIO	79.7
ENGINE LENGTH, in.	154
DIAMETER POWER HEAD, in.	100

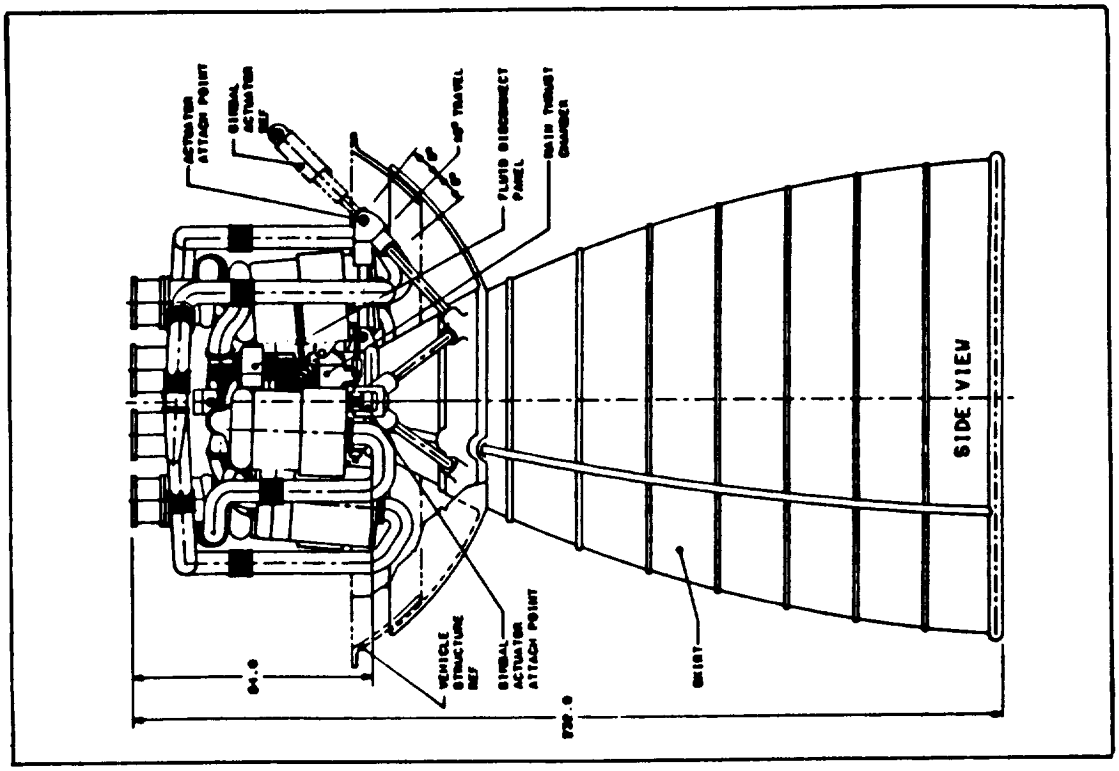

Figure 11: 1500K lb Thrust Booster/Sustainer Engine

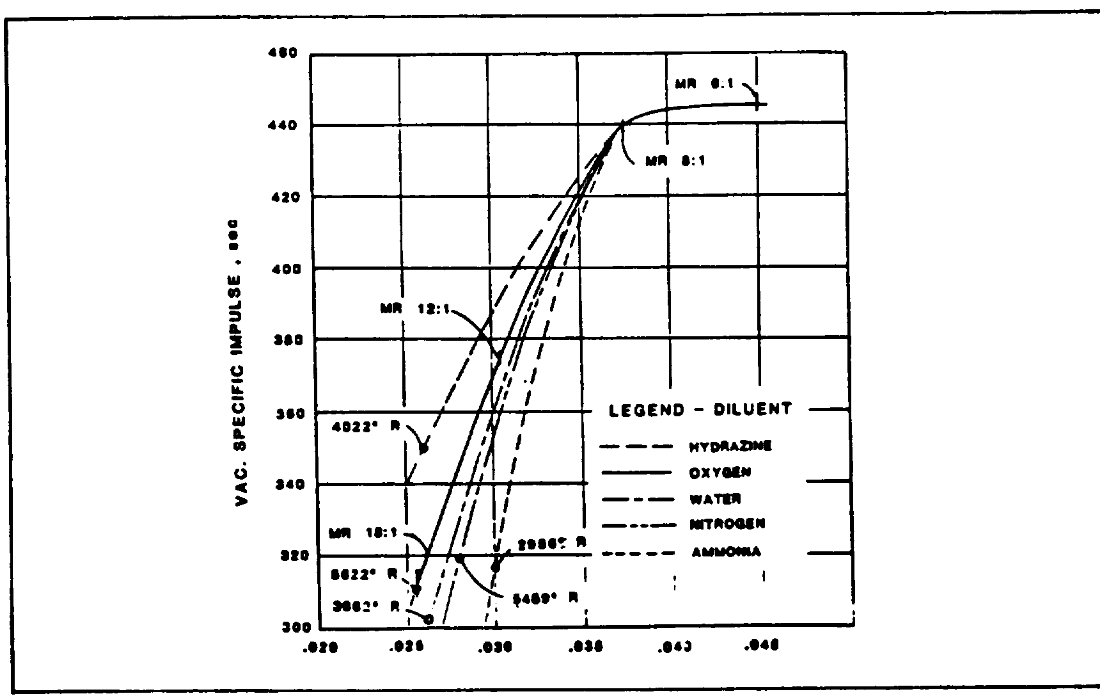

Figure 12: Specific Impulse Vs Bulk Specific Volume

This subject brings up an environmental issue. The rocket propulsion community essentially ignors the air pollution requirements which have been imposed on other fuel burning heat engines. The use of fuel-rich combustion is unlawful in motor vehicles, power plants, etc., in many loca-tions. We should take a hard look at what we are proposing to do with future launch vehicles which may have large use rates. In Florida the prevailing winds are off-shore. In California, the prevailing winds are on-shore hence, fuel-rich exhaust products are of concern. California counties such as Kern County (AFAL) have tough air pollution laws relative to hydrocarbon, carbon monoxide and oxides of nitrogen emissions. It would not be a big surprise to find that large scale testing/launching of fuel-rich LOX/-hydrocarbon booster rockets could not be done in some regions. These possibilities make the oxygen-rich LOX/LH_2 propellant combination look attractive for booster application.

NOZZLE SKIRT INSERT The optimum nozzle expansion area ratio varies significantly between sea level and the vacuum of space. Therefore, a means to provide near optimum area ratios is extremely important to the dry weight and gross liftoff weight for a given payload vehicle. Also there are large cost implications because the size of the rocket engines is significantly affected by the nozzle performance.

A two-position nozzle skirt is usually considered to be the most practical method of providing reasonable nozzle performance at low altitude and high altitude. One difficu-lty with the two-position nozzle is that the geometry of the rocket engine is such that the area ratio for low altitude operation is much too large (35-50) for best nozzle performance. The nozzle skirt insert or NSI depicted in Figure 13 overcomes the area ratio limitations of the translating two-position nozzle. The NSI is essentially a cylindrical member that is adapted to fit within the large diameter skirt of the nozzle. The NSI provides a low area ratio temporary flow path for the rocket engine exhaust gas stream during the early part of the booster phase of flight. Thus, the internal static pressure on the main nozzle skirt is substantially the same as the ambient pressure. As a result, the large main nozzle skirt does not cause pressure induced drag which reduces net engine thrust. The short period of service, i.e., about 60 seconds, required for the NSI allows the use of an ablative material such as carbon-carbon for construction. Thus the NSI does not require active cooling. After ejection of the NSI at a predetermined altitude, the gas flow fills the main nozzle skirt. This provides a high exit-to-throat area ratio which is desired for high altitude operation. The main nozzle skirt

may also be constructed from carbon-carbon or similar material having a low ablation rate so that active cooling of the large diameter portion of the skirt is not required.

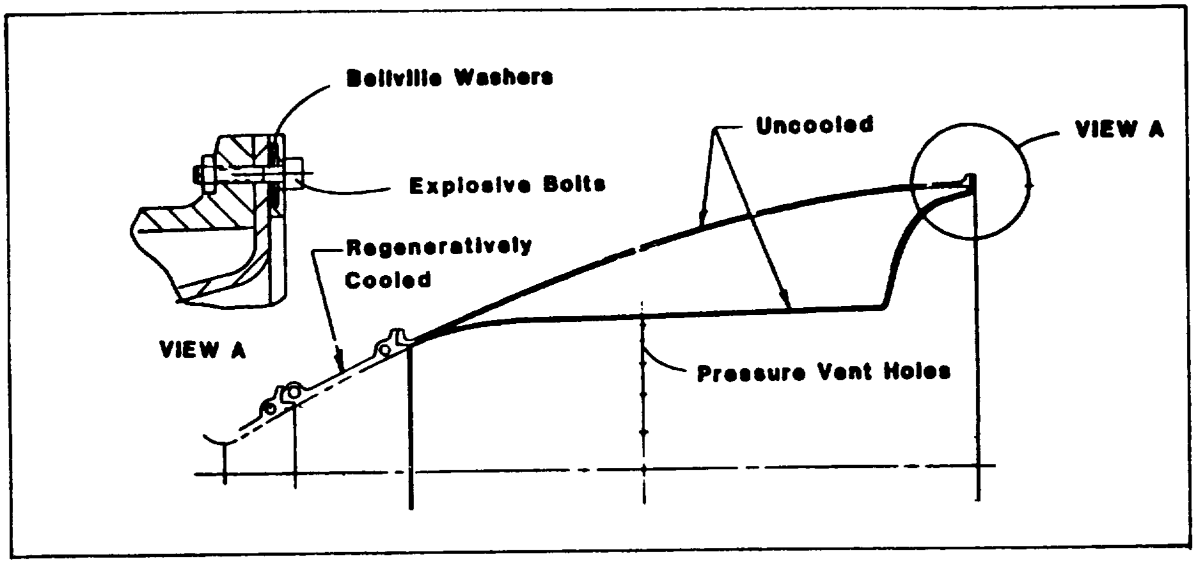

Figure 13: Nozzle Skirt Insert

The closing message for this paper is that for low cost rocket engines, the cycle selection should be toward engines which have operable and relatively independent sub-systems. Performance of the suggested engine type is insensitive to component efficiencies. The need for high hydrodynamic and thermodynamic efficiencies in order to achieve high Isp and modern chamber pressures is among the major cost drivers of today's propulsion systems. Conservative efficiencies typically permit large running clearances, large tolerances on parts, low temperatures, low stressses and high structural allowables for relatively inexpensive materials. These effects cascade through the entire engine program, paying dividends in high reliability at low cost. Cycles, such as the expander cycle, require an entire engine assembly for meaningful development testing. The gas generator cycle and the full-flow stage combustion cycle do not require an entire engine for component/sub-system development. If the rocket engine subsystems are separable, competitive procurement practices may be used for improved component/subsystem hardware at reduced cost. Also, when a new rocket engine is needed, the already developed subsystems may be used thus it will not be necessary to start from "scratch".